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Aspects of Aerodynamic Optimisation for Military Aircraft Design

B. Probert
Aerodynamics Department, British Aerospace
Military Aircraft & Aerostructures
Building W427D, Warton Aerodrome
Preston, Lancashire PR4 1AX, United Kingdom

1 Summary

The paper considers the role of various optimisation strategies in the aerodynamic design of military combat aircraft.

The multi – design point targets of military aircraft implies that the final product must achieve a carefully judged balance between, often conflicting, requirements. The current established way of working to achieve this 'balance' is first reviewed including the use of rule based procedures, the application of linearised CFD codes in both direct and inverse/optimisation modes, and the role of initial experimental data leading on to more detailed CFD work and experimental verification. Practical examples are given relating to the design of various projects including the Experimental Aircraft Programme (EAP), which was the forerunner of Eurofighter.

The need for improvements is identified, being primarily brought about by considerations of affordability and reduced design cycle time and also by the challenge posed from novel configurations to met low observability requirements. The means of achieving these improvements is discussed, and these imply the development of Multi Disciplinary Optimisation (MDO) in a wide sense. Numerical optimisation experience is reviewed but it is strongly emphasised that there is a need for rapid experimental input to the configuration design choice programme. Means of achieving this are discussed and examples given.

The high incidence requirements have a strong impact on CFD developments and areas of improvement are identified. This leads to a proposed new way of working implying a much stronger interaction between the initial and detailed design phases of aircraft design.

2 Introduction

The increasing emphasis on achieving processes that are more efficient and adopting concurrent engineering practices through the Integrated Product Team approach is producing dramatic changes in the way the design of an aircraft project is progressed. However before considering the need and potential scope for such changes it is worth reviewing the previous, and indeed current, ways of working and the achievements made using 'conventional' methodologies, since such methods still have much to offer.
The paper illustrates this by describing the aerodynamic design aspects of a number of different types of configuration. These differ according to their design requirements and are divided into three classes - transonic design emphasis, supersonic emphasis but with good transonic performance, and a supersonic dominant design. All of these share a common need for a rigorous interaction between the disciplines: aerodynamics / structures / design / stability & control (S&C) / systems / etc. but the main subject of this paper is aerodynamic optimisation. This covers shape optimisation for both performance and controllability, and the prospects for including this in a MDO environment.

In the context of wing design, it is recognised that the interaction between detailed shape optimisation for performance and structural optimisation is weak. The former usually optimises a camber / twist distribution for a given wing overall thickness distribution, whilst the latter depends dominantly on the thickness. The twist requirement will influence the wing structural optimisation but on military aircraft this requirement may be dominated by other considerations of controllability, in particular achieving high levels of roll rate at high speed.

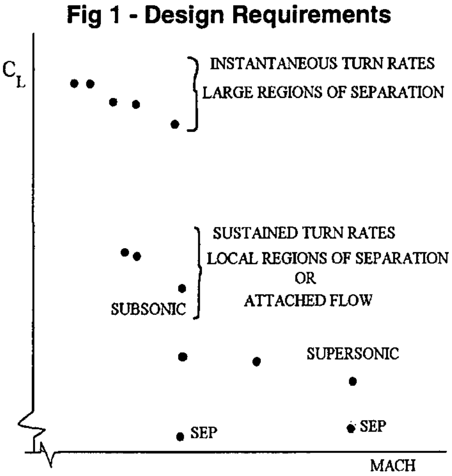
The wider optimisation issues for overall thickness, camber and twist design, particularly for supersonic performance has a strong link with structural optimisation and is an area which should benefit from MDO techniques.

Following the examples mentioned above the need for change is considered and means of achieving these is proposed. In this context, it is emphasised that not only increased automation and concurrency of existing methods are required but also there is a need to change the fundamental processes. It should be noted that the word 'change' is used, rather than 'improve', when considering an individual process. This reflects the increasing emphasis on affordability and reduction in design cycle time - not always compatible with quality improvements, though always adequate for the task in hand.

3 Requirements

The design of military aircraft is an extreme example of a multi-objective design problem. In the weapon system specification for performance requirements alone, there are usually a large number of point performances to be met along with many mission requirements. This is illustrated in fig. 1, which shows a typical set of requirements, viewed in the Mach Number - Lift Coefficient frame.

Fig 1 - Design Requirements



Thus, one cannot think of MDO as relating to the capture of all design requirements in one large design sequence. The development and application of MDO and optimisation systems will be sprinkled around different configuration issues and interfaces. In principle this will allow a better understanding of exchange rates and trade off studies and allow the consideration of a wider coverage of design parameters than achieved using conventional methods.

Paper presented at the RTO AVT Symposium on "Aerodynamic Design and Optimisation of Flight Vehicles in a Concurrent Multi-Disciplinary Environment", held in Ottawa, Canada, 18-21 October 1999, and published in RTO MP-35.

Even within the aerodynamic discipline itself there is a need to balance many related but conflicting requirements related to geometry / flight condition clashes. This is the case for an isolated wing - but in addition, there is a need to predict and allow for multi-component interference, e.g. wing - body interaction in the initial stages of design. Thus, it is usually not possible to think in terms of aerodynamic design for a wing in isolation. In addition, as the design progresses the full configuration will need to be evaluated using CFD and experimental facilities.

At the project feasibility stage there is a need, on a military project, to rapidly evaluate a large number of widely different layouts, in contrast to civil aircraft design where configuration changes tend to be perturbations of previous designs. This aspect has been compounded by the emphasis on achieving high standards of low observability leading to novel and non-ideal aerodynamic shapes, hence increasing the aerodynamic design challenge.

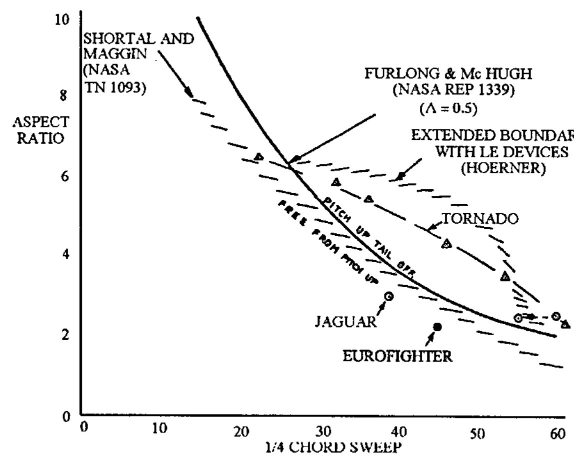
Referring to fig. 1 it can be seen that the major part of the flight envelope implies the presence of mild or severe regions of separated flow. Consequently, there is a need to predict both the onset of flow separation and the consequences on the configuration aerodynamics.

Finally, with all of the above in mind, there is a need to produce the best 'balance' in the configuration to meet the above objectives.

4 Current means of achieving requirements

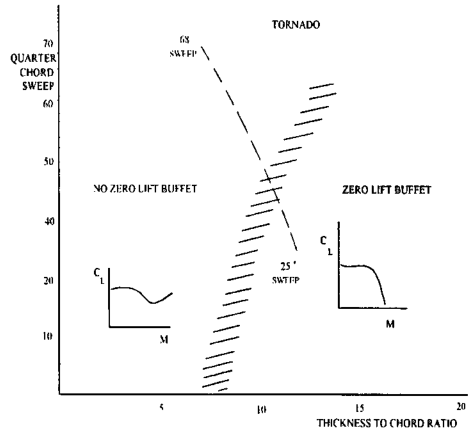
Considering the aerodynamic aspects of design optimisation, the means of achieving requirements are illustrated via a number of examples. In the initial stages of design the skeleton layout, sizing, initial fuselage shaping, packing, wing planform, etc. will be done using a 'rule based' approach founded on previous experience and a host of empirical methods. For the wing planform and thickness selection, reference to guidelines as shown in figs 2 and 3 may be made.

Fig 2 - Pitch up Boundaries



The first figure gives a relationship between wing aspect ratio and wing sweep that should ensure that the configuration avoids pitch up. However the empirical correlation is based on conventional trapezoidal planforms and its validity to more novel geometries is debatable. The latter figure shows a correlation between wing thickness and wing sweep which relates to the onset of severe buffet / shock stall, again based on trapezoidal planforms with the same doubts about general applicability. The above two examples of simple rules in preliminary design will, if applied, lead to a configuration that needs thorough evaluation and that will usually need substantial enhancement to achieve an adequate design standard.

Fig 3 - Combination of Sweep and Thickness

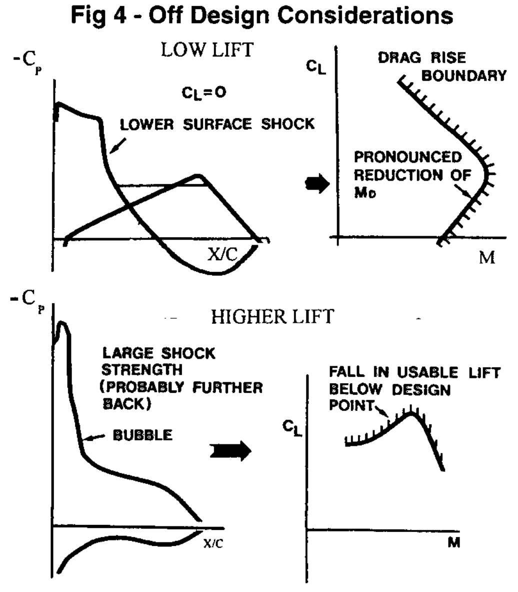


4.1 Transonic design case.

4.1.1 Thin Wing - supersonic performance as 'fallout'

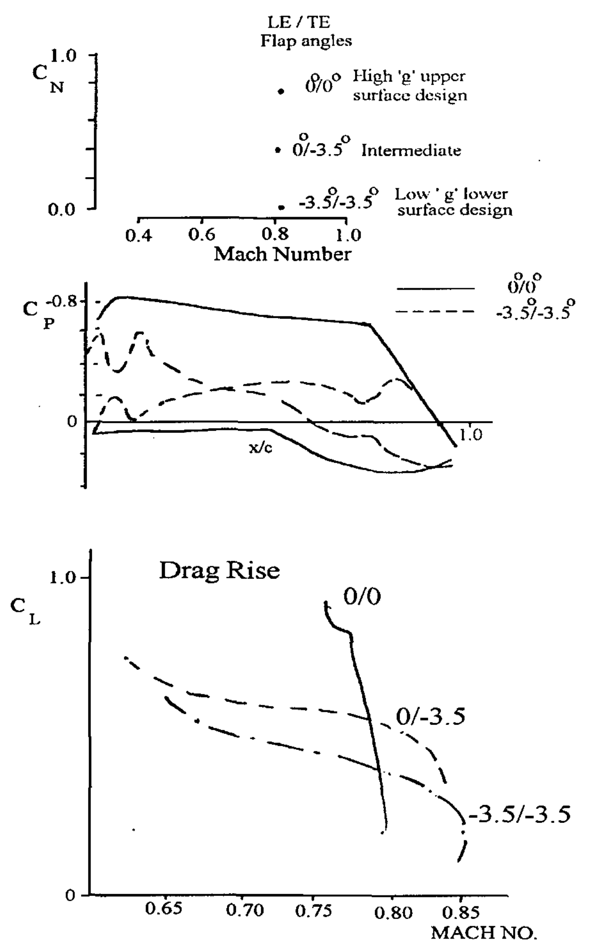
The first example illustrates optimisation for a configuration with outstanding transonic flight performance using supercritical wing technology allied with variable camber at the leading and trailing edge (Refs 1, 2).
The configuration consisted of a high wing layout, with optional 'non interfering strake' to improve high incidence penetration with little detriment to low incidence drag. The wing LE sweep was 42° with a conventional trapezoidal planform. However, the initial wing design task was tackled by the design of aerodynamically equivalent 2D wing sections. The problem was to derive a section to meet at least two design points at high subsonic Mach number. The first was a high 'g' 'sustained turn rate' (STR) point and the second a low altitude high-speed dash point. If a fixed geometry is designed for the first condition then this will lead to separation on the lower surface for the second design condition. This leads to a pronounced reduction in drag rise Mach number at low lift as shown in fig. 4. (This figure also shows potential problems at high lift that can be improved, obviously, by additional LE flap deflection).

Fig 4 - Off Design Considerations



Use of variable camber solves the lower surface problem; the LE and TE flap are deflected upwards by 3.5° as indicated in fig. 5. This figure also indicates the large extent of supercritical flow achieved at the STR condition leading to a high value of lift to drag ratio.

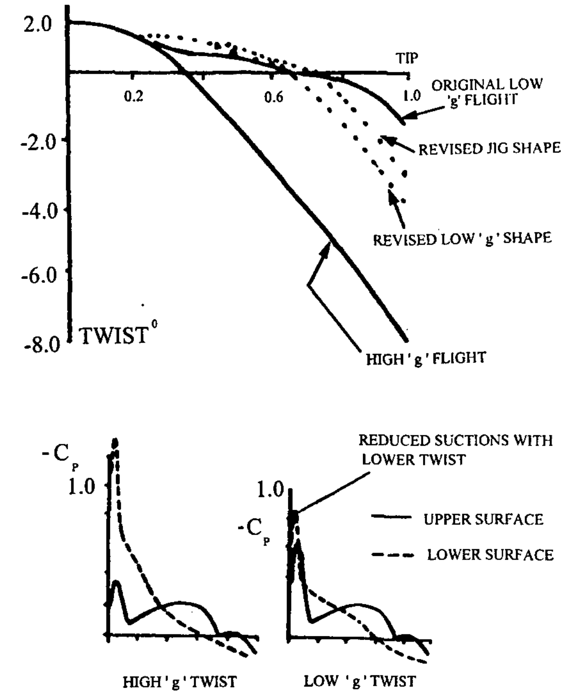
Fig 5 - Benefits of Variable Camber Theoretical Design Equivalent 2D Aerofoil Section



The resulting drag rise boundaries are shown in the lower part of fig. 5 where the extended penetration to higher speeds at low lift is evident.
This design was achieved through an iterative application of CFD codes designing the upper and lower surfaces separately at each design point and through the design and control of the surface curvature distributions. The 2D, and later the 3D design again adopting a repeated application of CFD codes, produced excellent standards of performance. The resulting 3D geometric shape for the STR point was deemed to represent the in-flight geometry and so included the effects of aeroelasticity. This was removed via an iterative procedure in order to derive the wing 'jig' shape for possible manufacture.

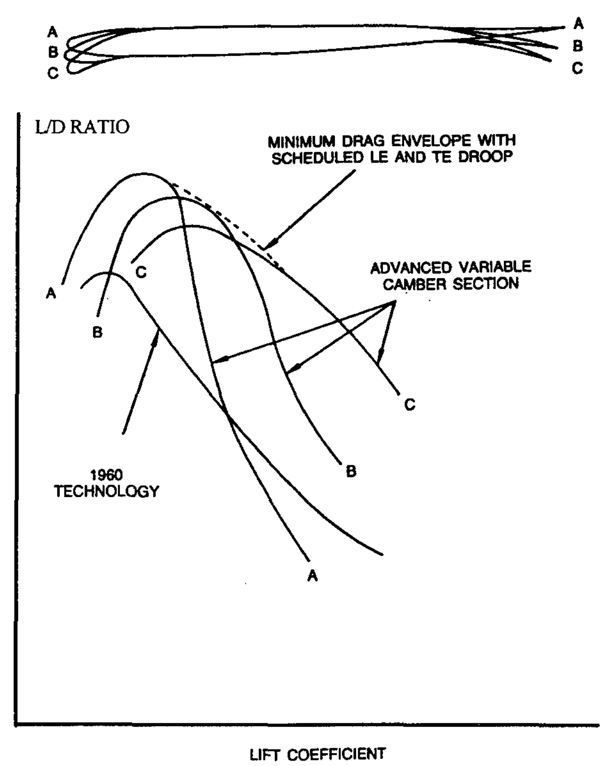
The flight and 'jig' twist values are compared in fig. 6. It was observed that 60% of the aeroelastic twist was obtained naturally through the wing bending contribution assuming a conventional aluminium alloy skinned multi-spar structure. The remaining 'jig' twist, when coupled with up LE flap gave satisfactory performance at the low lift design point.

Fig 6 - Twist and Aeroelastic Effects



The final result, compared with a 1960's technology configuration, is illustrated in fig. 7 and shows excellent performance gains. It is not claimed that such a design represents the optimum solution for the problem but significant advances were made over previous design standards.

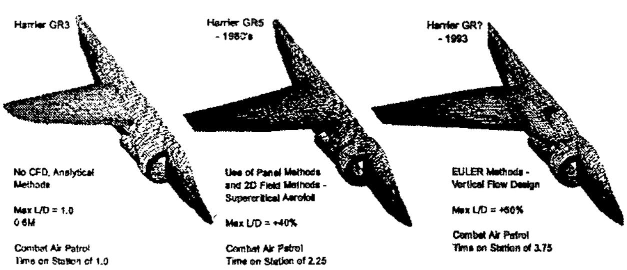
Fig 7 - Effect of Advanced Variable Camber Wing Design on Lift/Drag Ratio



4.1.2 Thick wing transonic design - subsonic aircraft

Here illustrated by the progress in design applications of CFD methods for the Harrier VTOL configuration wing design - as shown in fig. 8. The application of supercritical aerofoil technology and improvements in CFD methodology, including the incorporation of a vortical flow design, produced progressive increases in performance. These were achieved through repetitive CFD and geometry manipulation packages and led to 50% gains in maximum Lift to Drag ratio.

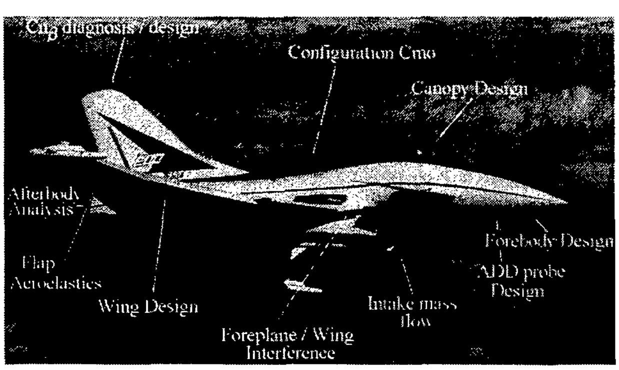
Fig 8 - Progress in Aircraft Wing Design CFD's Traditional Application



4.2 Supersonic bias design

The second example considers a conventional design approach as applied to the Experimental Aircraft Programme, a cranked delta with foreplane layout, which was the forerunner to Eurofighter. Here the supersonic requirement was severe, demanding a 'Beyond Visual Range' (BVR) capability at Mach 1.6 and requiring excellent transonic performance as well. Fig. 9 shows the configuration and the areas where CFD played a part in the overall optimisation. Here we concentrate only on the wing related optimisation.

Fig 9 - EAP: Areas of CFD Application - Canard Delta



At the outset it was realised that the canard - delta configuration was a significant departure from previous experience and the first step was to carry out a large low speed experimental programme on a 1/10th scale model aimed at optimising the wing planform and wing to foreplane relationship. This study was outside the capability of CFD and configuration choice was strongly influenced by the S&C characteristics, dominantly at high incidence, including the ability to recover from high incidence. The experimental database allowed exchange rates to be quantified between S & C and performance (maximum lift) parameters. Example exchange rates are shown in figures 10 and 11 and these guided the final choice of planform.

Fig 10 - Use of Experimental Data in Configuration Design - Maximum Lift Trends

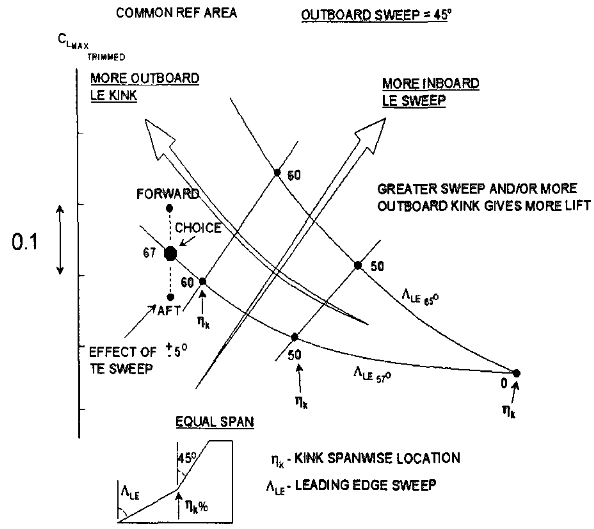
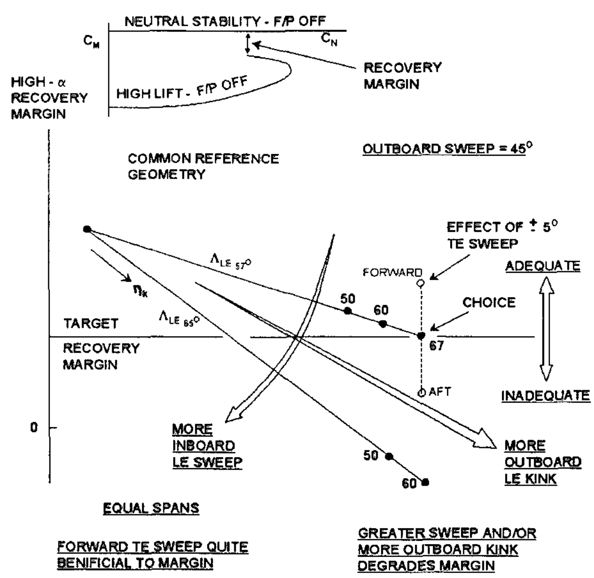


Fig 11 - Use of Experimental Data in Configuration Design - Recovery from High Incidence



The best CFD code available at the time was an EULER code capable of treating a wing placed on a simple body composed of circular cross sections. This code gave good predictions of the wing pressure distributions and total lift at both sub and supersonic Mach numbers but poor prediction of the overall longitudinal pitching moment. To improve matters a hybrid procedure was derived which combined the strength of a potential flow panel method, to represent complex wing-fuselage geometry, with the non-linear wing flow capability of the EULER method. The resulting procedure gave accurate predictions of pitching moment for both stability and zero lift moment as shown in fig. 12.

The prediction of both these aerodynamic quantities are fundamental to achieve a balanced design as they affect controllability and efficiency. The new procedure also highlighted the complex nature of the wing to body interference for this configuration, which is shown in fig. 13 for the variation of zero lift moment with Mach number. The body on wing and wing on body effect increases dramatically at transonic speeds though the wing alone trend is benign. The procedure also agrees well with the experimental data shown as the filled in symbols and was thus used to design and assess later developments.

Fig 12 - Comparison - Modified Euler v Experiment
M = 0.9

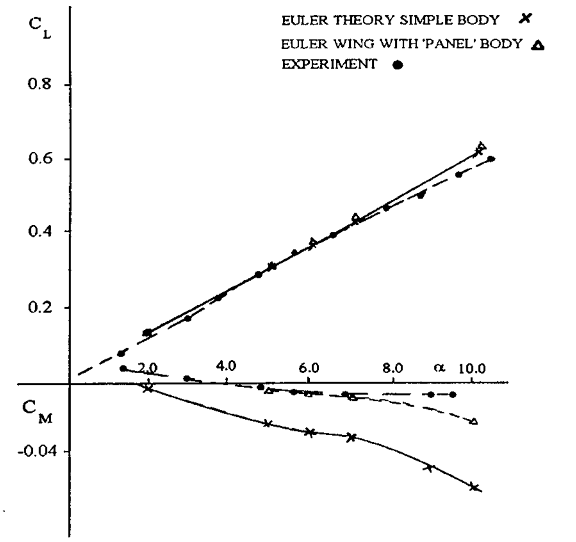
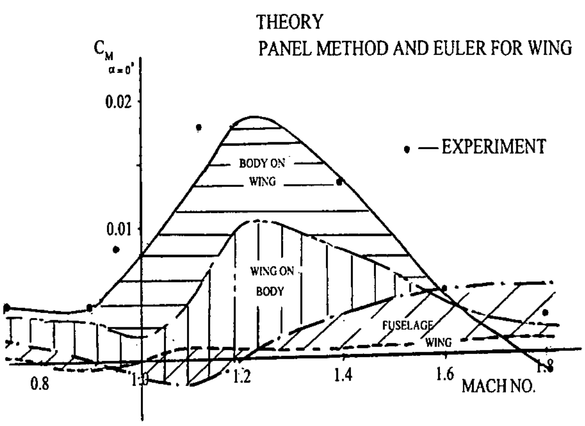


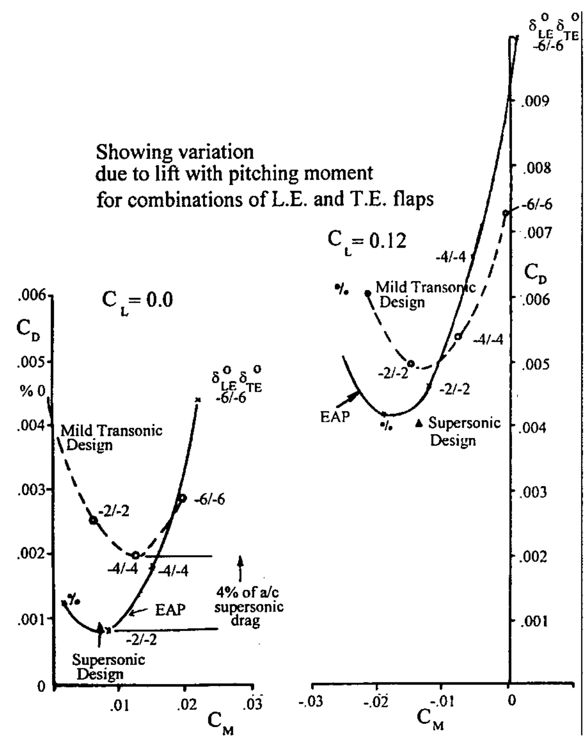
Fig 13 - Pitching Moment at Zero Incidence



This is an example where the designer makes use of, at the time, an inadequate CFD capability but improvises to produce an effective design procedure. Had an automatic optimisation capability been available at the time it would not have been successful since use of the basic CFD alone would not have been adequate.

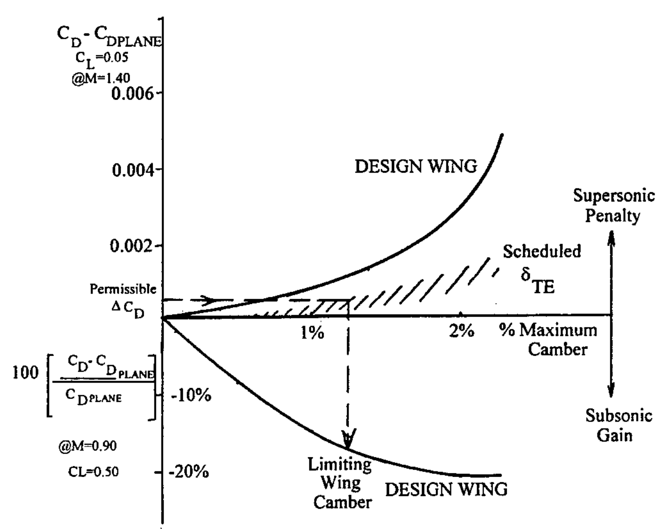
The search for a configuration with excellent supersonic performance coupled with very good transonic/subsonic performance is difficult. It was found that the best balance was achieved by designing for the supersonic manoeuvre point using a combination of linear theory and an EULER code and achieving the sub/transonic performance with LE and TE flap. The alternative of designing for the transonic case and decambering to achieve the supersonic case was not as successful. This is illustrated in fig. 14 for the early work done on the EAP at a Mach number of 1.4. The predicted drag values (via EULER) are plotted against pitching moment at two constant values of lift coefficient, effectively allowing comparison at trimmed conditions. The variation in drag with various LE/TE deflections is also shown and it is clear that even a mild transonic design suffers a 4% penalty in zero lift drag compared with a supersonic design case. The final design selected was based on a refinement of a supersonic linearised theory optimisation.

Fig 14- Supersonic Drag - Lifting Surface Theory
M = 1.4 Wing + Body



By a process of repetitive manual design using CFD an exchange rate was derived between wing maximum camber levels, subsonic STR performance and supersonic '1g' performance as shown in fig. 15. This served as a useful guide in checking permissible camber levels.

Fig 15 - Exchange Rate



The final 'balanced' design gave large gains in sub/transonic performance with only a small penalty to supersonic cruise performance as shown in fig. 16.

In parallel with the shape optimisation work the conventional aerodynamic loading / structural / aeroelastic / flutter cycle was proceeding. In this instance the design and manufacture of the 'loads' fully pressure plotted large scale (1/10[th]) high speed tunnel model and subsequent testing covered a period of approximately two years from the start date. The information from the model was not available until after first flight and so it did not contribute to a better standard of loading estimate pre flight, though it did contribute to the check stress and flight clearance activities. The time expended and the cost of the model was significant.

Fig 16 - Benefits of Twist and Camber

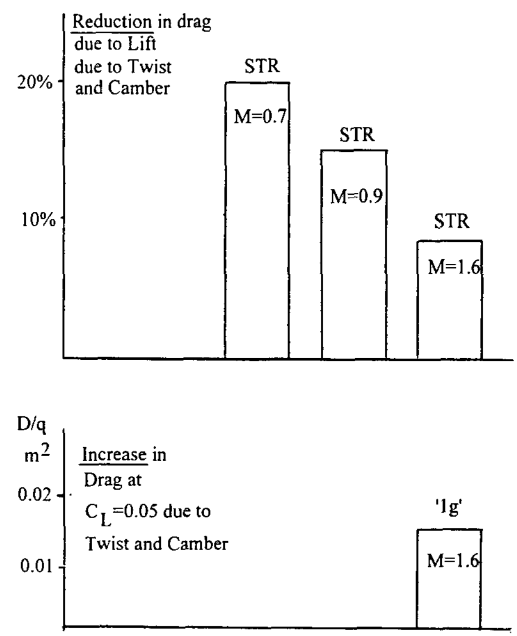
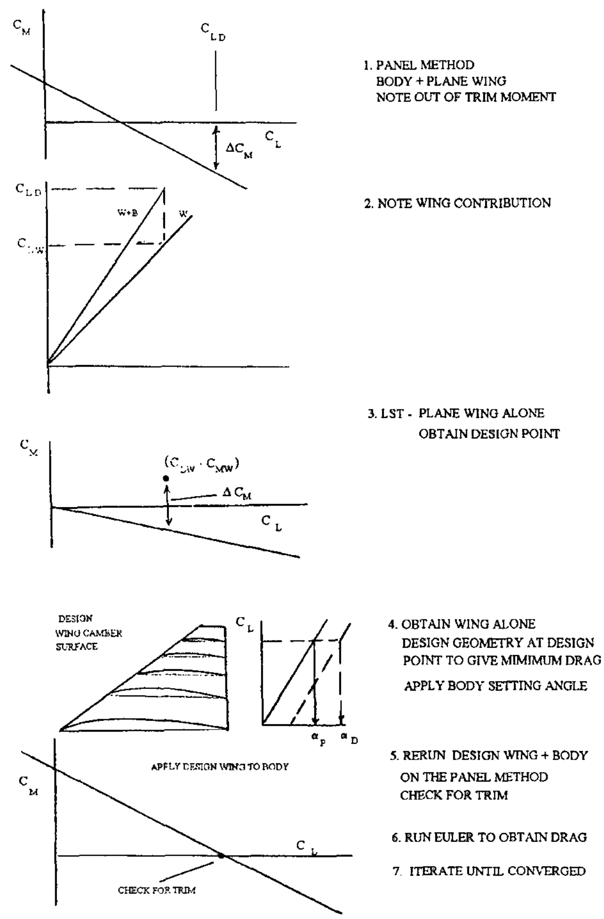


Fig 17 - Hybrid Supersonic Design Procedure



4.3 Supersonic design

The third example illustrates the design of a supersonic configuration. The second example above has already highlighted the importance of trim drag at supersonic speeds and to allow for this a procedure was developed to optimise the wing, using linearised theory, in the presence of the complex fuselage flow field derived from a panel method with the aim of minimising drag at trimmed conditions. The process is outlined in fig. 17. At each iteration the whole configuration is run on a panel method to check the 'out of trim' increment. Corrections are obtained and the process repeated until satisfactory convergence is achieved. Less than six iterations are required to obtain a satisfactory result. The final drag level is obtained using a 3D EULER code The above is an example of numerical optimisation using linear theory allowing for external interference. The process involves a 'man in the loop' due to only partial automation and the need to inspect intermediate results.

An example of optimisation using non-linear (EULER) theory (Refs 3, 4) is given in fig. 18(a/b) where a wing geometry is improved using the cross flow shock 'analogy' with 2D aerofoil design concepts. The need is essentially to increase L/D at supersonic manoeuvre conditions. The basic idea is to view the flow in a direction normal to the shock wave and to modify the wing geometry so as to weaken this shock wave. Such an exercise was undertaken (Ref. 5) and successfully reduced the drag but also showed the importance of allowing for trim drag at supersonic speeds - see fig. 19. Here trimming is done using the trailing edge flap.

Fig 18(a) - A Supersonic Design Approach

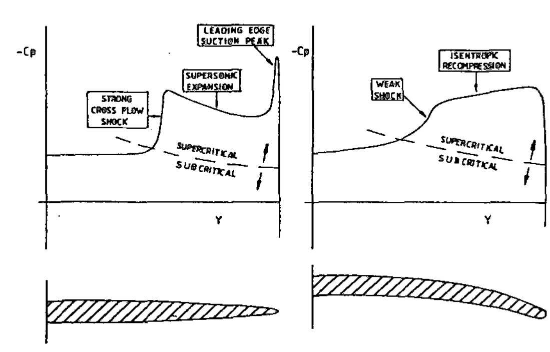
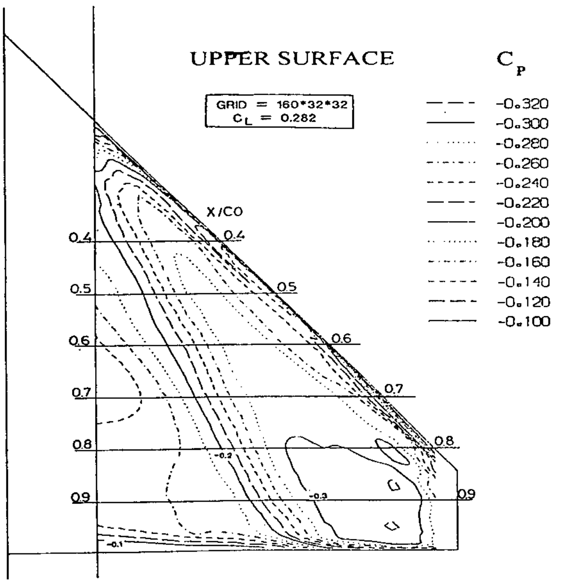
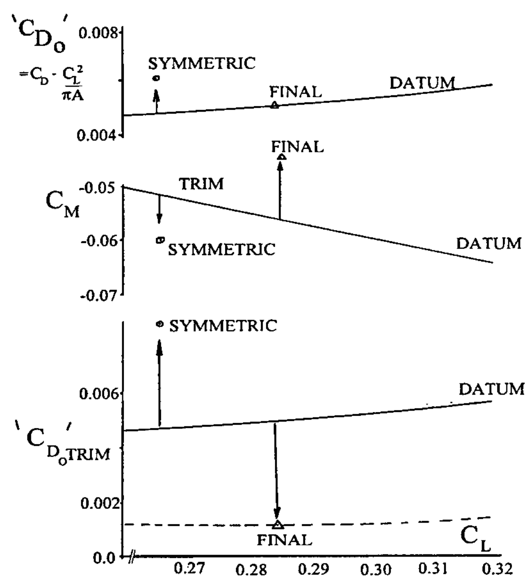


Fig 18(b) - A Supersonic Design Approach



WING ISOBAR PATTERN AT THE SUPERSONIC MANOEUVRE DESIGN POINT: M=1.6, n=4'g' FINE GRID

Fig 19 - Importance of Trim Drag



Three designs are shown - a datum wing, a symmetric wing and an improved wing at the same design lift coefficient. The latter and the datum have the same untrimmed drag, but its more positive pitching moment leads to a large reduction in overall drag when trimmed. This effect can be larger than the gain achieved by the 'cross flow' technique mentioned above and implies that a pitching moment constraint should always be used in supersonic optimisation. Again this optimisation was achieved using 'ad hoc' algebraic geometry modifications in both chordwise and spanwise directions, with manual intermediate inspection/decision making and repetitive use of CFD.

5 The need for improvements

The examples above have highlighted a number of tasks that are essential to the design process, but are time consuming to perform and could probably be done in a better way.

The first and the third design examples involved repetitive 'man in the loop' calculations and geometry manipulation to solve a multi-point performance problem. This is obviously a prime candidate for numerical optimisation but only since the design involved STR and low lift conditions where the design aim was to achieve attached flow (where the CFD method is valid). Extending this work to include 'instantaneous turn rate' (ITR) requirements at high incidence implies the ability to predict separated flow development. It is also evident that potential current low-observability requirements imply non-ideal aerodynamic options, where the onset of flow separations can begin at lower incidences than usual. The consequences can also be more severe. Hence, there is a need to include improved CFD capabilities in design optimisation both in terms of the flow physics modelling and to improve response times.

The second example showed that planform choice and wing to foreplane relationship was selected mainly on the results of wind tunnel data at high incidences and so was the first step in the detailed optimisation work which followed. The related loading / structural optimisation also covered a long elapsed time mainly due to the need for extensive pressure plot confirmation.

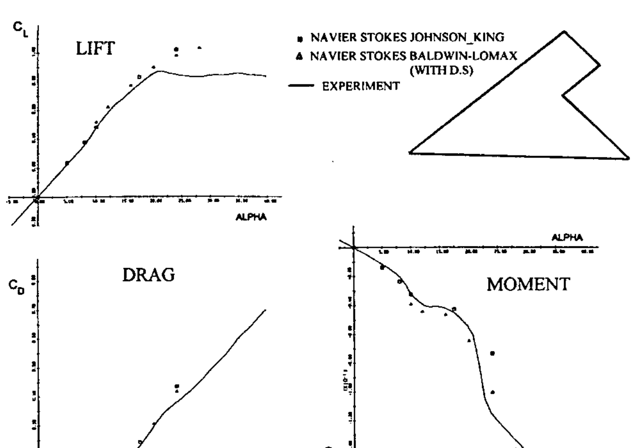
There are thus many areas for improvement.

6 Possible developments

6.1 CFD enhancements

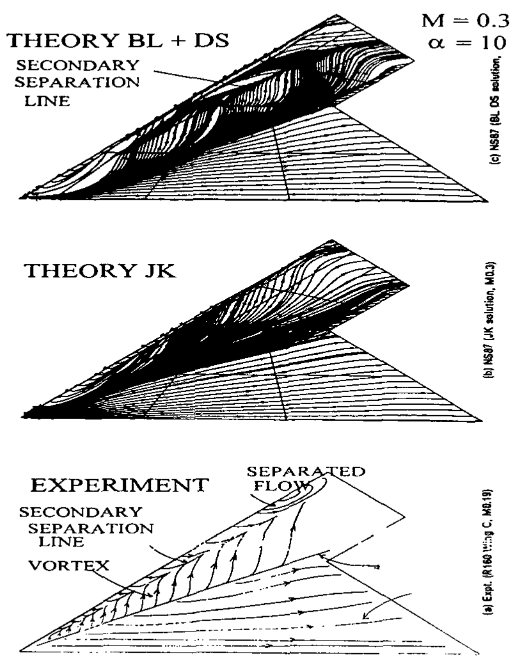
Recent work on the application of a Navier-Stokes code to the flow over a generic novel geometry, where flow separation occurs early, has shown promise as shown in fig. 20 below.

Fig 20 - Navier-Stokes Comparison with Experiment



The method was run on the gross wing geometry alone and the results compared with low speed experimental data on a wing-body configuration. The body was narrow and of a simple nature with constant elliptic cross section and did not adversely interfere with the wing flow. The predictions agree well with the experimental force results up to high incidence. Maximum lift is not predicted but pitching moment and drag trends are well predicted. A more detailed look at the predicted skin friction lines on the wing upper surface is also compared with experiment in fig. 21. The experimental result lies in between the Navier Stokes predictions, shown for two turbulence models.

Fig 21 - Novel Wings
Theory (Navier-Stokes). v. Experiment
- Surface Friction Lines



However current configurations exhibit more complex wing / body / chine / empennage geometries. The aerodynamics of these, on current evidence, are unlikely to be predicted adequately through the use of CFD at high incidences, though limited regions of separation can be qualitatively predicted. There is thus the limited possibility of introducing a Navier-Stokes capability into the optimisation problem but the computing requirements would almost certainly be prohibitive. Thus the use of such a method is likely to be intermittent, providing a 'physics reality' check when embedded in a design optimisation loop involving a lower order CFD code (e.g. EULER) - essentially a form of multi-level optimisation.

The inadequacy of CFD for treating complex configurations at high incidence places increased emphasis on experimental work, which formed an essential part of the second example in section 4.2 above.

6.2 Experimental developments

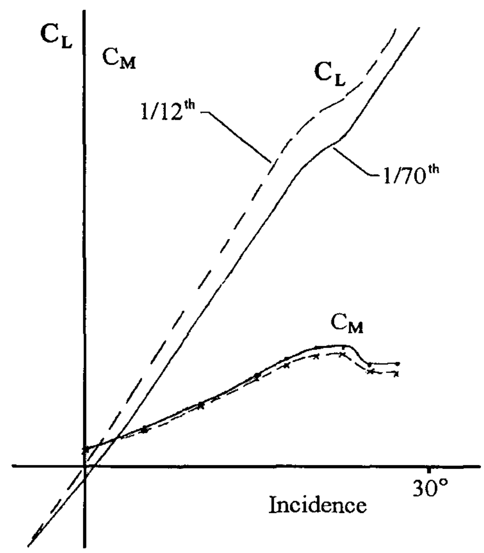
6.2.1 Simplified Experimental Models and Test

The approach adopted at BAe Military Aircraft & Aerostructures (MA&A) has been to increase productivity through the use of simplified and smaller wind tunnel models (1/70th scale), which has also enabled the use of smaller and simpler low speed tunnels, e.g. a small open jet facility.

Models are constructed with a 'straight through' constant thickness 1mm flat plate metal wing to which the fuselage components are attached, with the empennage attached to the fuselage or wing as necessary.

There is no profile shaping and no LE shaping so the primary interest is simply to gain rapid information on a number of alternative body-chine-wing layouts. This technique has been successfully employed at MA&A after an evaluation of the procedure was made against conventionally gathered test data using a larger model (1/12th scale) and tunnel. A comparison of the two series of test data is shown in fig. 22 where it is seen that although an exact match is not produced, the trends and force changes occur at similar incidences so the data can be used to give relative comparisons between different configurations. It has also been demonstrated that the incremental effects of control surfaces are reliably predicted.

Fig 22 - Comparison of Small and Large Model Data
Lift and Moment



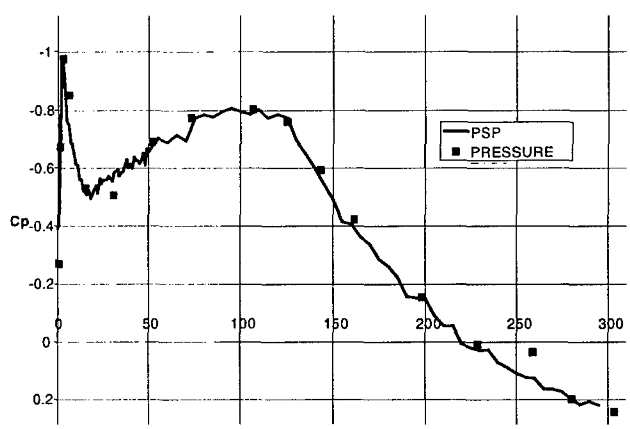
The savings in cost, design effort and in particular elapsed time are considerable. However the flow type on all the configurations tested in this development study was vortical in nature and thus not sensitive to scale effect or to wing geometry (for a model with a thin, but blunt, LE). It is an example where such an approach may be 'fit for purpose' but it can not be regarded as a generally applicable technique - especially for wings of lower sweep. However the rapid and extensive information gained with this approach allowed planform and configuration initial selection to be made on high incidence and control power characteristics.

6.2.2 Loading issues and Pressure Sensitive Paint

As noted earlier in section 4.2 the detailed surface pressure data was obtained at a very late stage in the design cycle, placing a heavy responsibility on the earlier estimates obtained based on past experience, related experimental data and CFD codes. This is especially true on novel configurations where past experience and databases may be inadequate and CFD is not well evaluated. The latter is also true for intended CFD shape optimisation work where early CFD evaluation is essential.

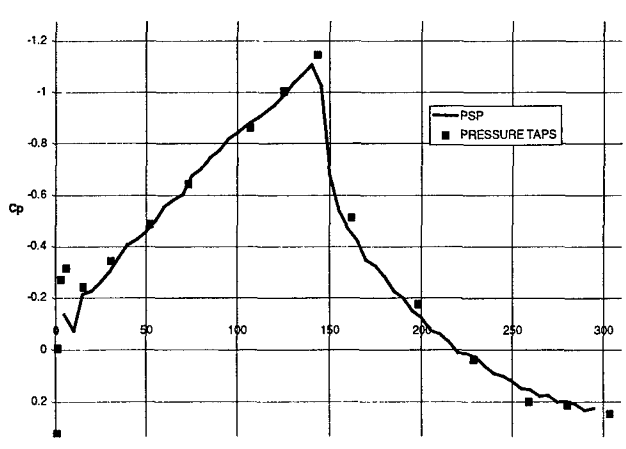
At BAe (MA&A) the current view is that pressure sensitive paint offers the best means of obtaining rapid and cost effective pressure data. Such a system (SUPREMO) (Ref. 6) is under development and several trials have taken place. The BAe system is based on a ' time decay' signal analysis rather than the usual luminous intensity based systems. In principle, the method is more accurate at and more tolerant of low intensity levels and high surface curvature regions. Typical results obtained on a 12 inch chord aerofoil model spanning the BAe (Warton) 1.2m HSWT are shown in fig. 23 and fig. 24 where comparison with experimental pressure tapping data is excellent. The detector was mounted so as to predict the development of the lower surface pressures, which accounts for the unusual pressure distributions.

Fig 23 - PSP Comparison with Pressure Tappings 2D Section Mach 0.69 Incidence -4°



Both LE suction peak and the general chordwise development are well predicted though the position of the shock wave is slightly different in fig. 24. The possibility of a paint intrusion effect or a slightly non-2D development on the model is under investigation. In addition the method is being developed and assessed for 3D configurations.

Fig 24 - PSP Comparison with Pressure Tappings - with shock Mach 0.77 Incidence -2°



The technique offers reduced risks both in terms of loading and design optimisation work, reduced timescales, reduced model costs through the elimination of the 'loads' model, and continuous loading and design checks as the configuration evolves through the ability to 'piggy-back' conventional S & C testing.

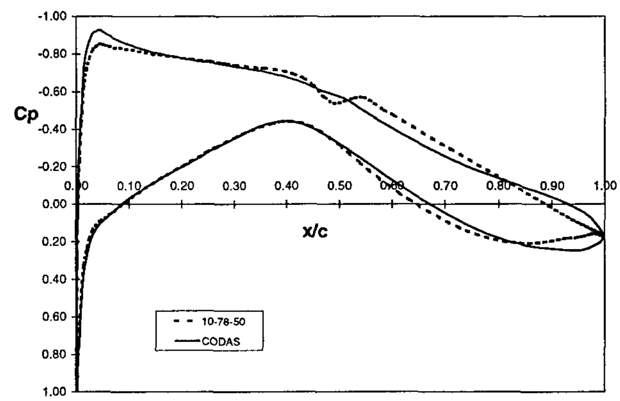
6.3 Numerical Optimisation

A method which is receiving significant attention and evaluation at BAe is the shape optimisation system developed by DERA called CODAS (e.g. Ref. 7). This is a constrained non-linear gradient based optimisation code linked with a curvature based incremental geometry capability. It can be linked with a number of CFD codes, potential flow or Euler based codes in both 2D and 3D. Impressive evaluations and applications have been undertaken on both 2D and 3D wing designs though it must be stressed that such methods are to be considered as another 'tool' in the designers kit as great care is needed to drive the method to a proper solution.

Typical evaluations investigated the choice and number of design variables and their scaling, alternative formulations of objective and constraint setting, and considered both single and multi point designs. A resulting pressure distribution from the method is compared with a good datum standard of design in fig. 25 where a performance achievement is obtained through smoothing of an already weak upper surface shock and a modification of the rear surface loading.

Fig 25 - 2D Section Design

10-78-50 Aerofoil v 2D CODAS Optimisation



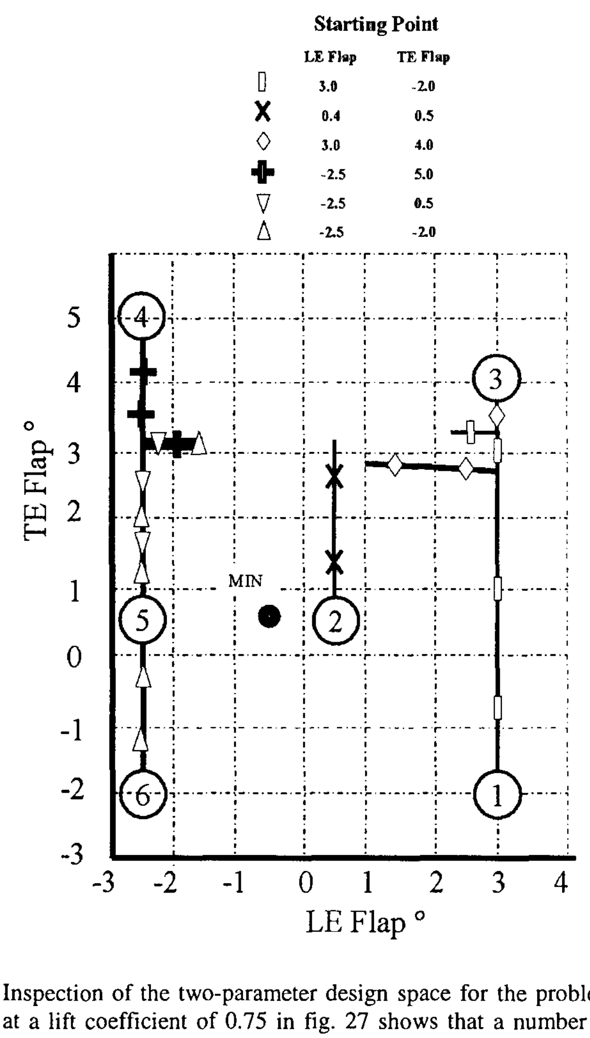
Further examples are too numerous to mention but the general conclusions reached were that the method worked best when adequately constrained, thus avoiding local minima, and that the effect of the design variable scaling should be investigated early on. For a 3D case this is difficult due to the computing requirements involved. However CODAS suggested options that may well have been dismissed using conventional approaches and it was noted that drag minimisation alone is not a sufficient objective, some form of control on the pressures is also required along with physical constraints to guide and control the optimisation process. Further evaluations and applications are planned.

The comments made above are reflected in a related but more extensive study undertaken by ESDU (Ref. 8). The same code was applied and evaluated for a range of 2D cases optimising section camber, upper and lower surface geometry independently and LE/TE flap schedules. The results illustrate the care that needs to be taken with gradient based optimisers. This is shown for an example where the objective is to minimise drag, at a design lift coefficient of 0.75, with no other constraint applied, by the deflection of optimum settings of LE and TE flap.

Fig. 26 shows the solution paths starting from five different starting points (i.e. values of LE/TE deflection) for a case where the incidence is started at a low value of 1°. The actual minimum drag point is marked and it is clear that the process does not approach the correct solution from any of the starting points. Subsequent investigation of the results showed that the solution paths are strongly dependent on the scaling factors applied to the design and the fact that the initial starting point, in each case, is well outside the 'feasible' region (here, of lift coefficient). The method immediately attempts to reach the feasible region by changing the design variable that gives the largest increment towards that end. This increment is also a function of the input scaling factors.

In fig. 26 the scaling factors used are larger for the flap deflections than for incidence and so the method rapidly increases the former to reach the 'feasible region', but not an optimum.

Fig 26 - Effect of Different Starting Points on Design Variable Paths
Scaling Factors LE/TE/ALPHA (8/8/1)



Inspection of the two-parameter design space for the problem at a lift coefficient of 0.75 in fig. 27 shows that a number of solutions are possible with the gradient based approach. If the scaling factors, in particular, are chosen appropriately one would expect this behaviour.

Revising the scaling factor on incidence does indicate the expected behaviour for four of the cases, as shown in fig. 28. Thus it is essential to try different starting points and to investigate such influences early on especially where a large number of design variables are employed where a simple 'visualisation' of the solution, as done above, may be impractical.

Fig 27 - Anticipated Effect of Different Starting Points on Solution Paths

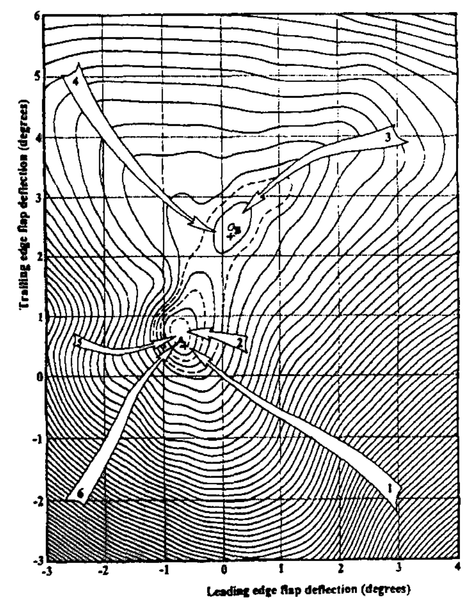
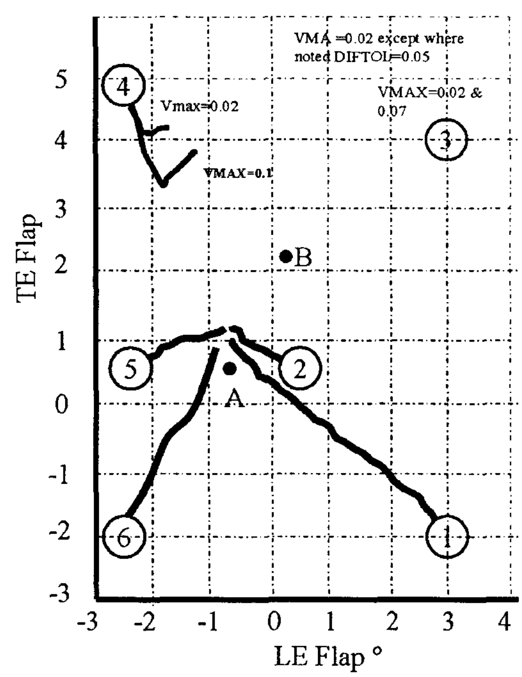


Fig 28 - Revised Scaling Factors LE/TE/ALPHA (8/8/5)
Effect on calculation paths



The option of treating this problem using a different (e.g. stochastic) optimisation procedure is yet to be addressed.

6.4 Multi-Disciplinary Optimisation

A system currently under evaluation at MA&A (Warton) is FRONTIER (Ref. 9), produced as a result of the FRONTIER European Framework IV project. This is essentially a framework that allows distributed processing using any user-required executables. Alternative optimisation packages are also available, including a genetic algorithm option. The system can be used to produce the Pareto boundary for key performance parameters and has already been applied by the

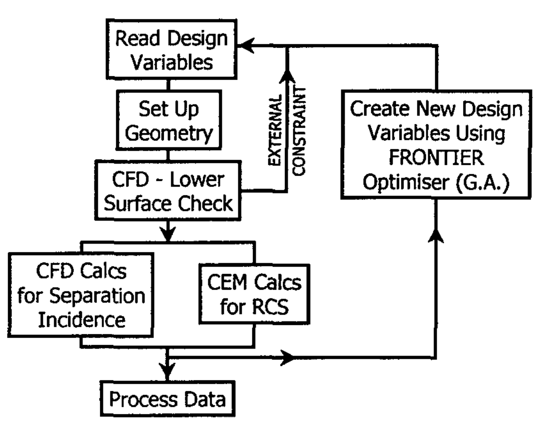
partner companies/developers to a number of design problems. Application by DERA and BAe to the optimisation of a thick delta wing has been reported in Ref. 10 while a more recent BAe application to a cavity / store release problem is given in this symposium as paper no. 14 (Ref. 11).
A further application is described below where the two disciplines involved are aerodynamics and radar signature. The case is simple in nature and attempts to assess the effect of aerofoil leading edge geometry on both the aerodynamics and Radar Cross Section, and whether a Pareto boundary exists for this type of interchange.

The aerodynamic problem posed is to assess the **off design** effect of leading edge changes by estimating the low Mach number ($M = 0.50$) upper surface flow separation boundary - thus the incidence at which separation starts is to be maximised. As a prerequisite, at low incidence (0°) a check is done to ensure that suction peaks on the lower surface also conform to a prescribed suction limit so that a 'high speed dash' can be achieved at low altitude.

The RCS problem posed is that of an aircraft, flying at low altitude, approaching a ground based radar operating at 3GHz. Thus incident elevation angles are small (0.5° to 1.5°). The RCS is averaged over this incidence range.

The problem is reduced to only three geometric design variables - the **leading edge radius** and the **two angles** which define the upper and lower surface blend point between the LE radius and the rest of the aerofoil. The latter is taken to be an elliptic profile back to the aerofoil maximum thickness on both surfaces. As these three variables are changed the leading edge droop or camber changes along with the curvature distribution. The flow process chart is shown in fig. 29. An Euler code has been used for the CFD calculations and a limiting pressure coefficient applied to derive the separation boundary.
The RCS results were derived using a frequency domain code which solves Helmholtz' equation.

Fig 29 - Process Flow Chart



The first results obtained, where the allowable range of each design variable was restricted are shown in fig. 30. A clear demarcation boundary is seen, though the number of cases is not large. Analysis of this data and additional runs indicated definite trends; also shown in fig. 30. It is interesting to note that an increase in nose radius affects both separation incidence and the RCS return, assisting the first and penalising the latter, as expected. However increasing Θupper dominantly affects the separation incidence by increasing the LE 'droop'. Increasing Θlower dominantly affects the RCS. Both of these are more subtle effects and can be used to counter or balance a LE radius change.

Fig 30 – Narrow Range Variables and Trends

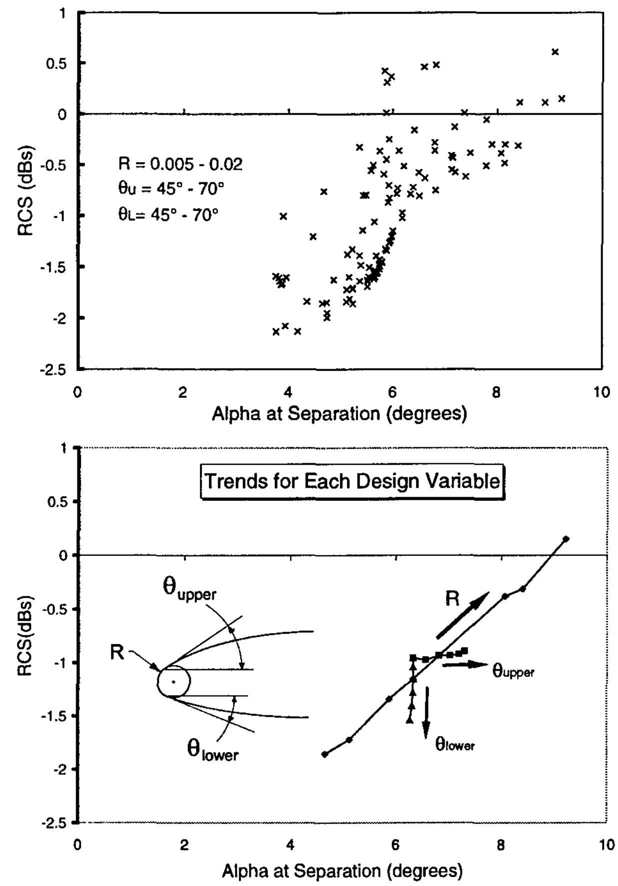
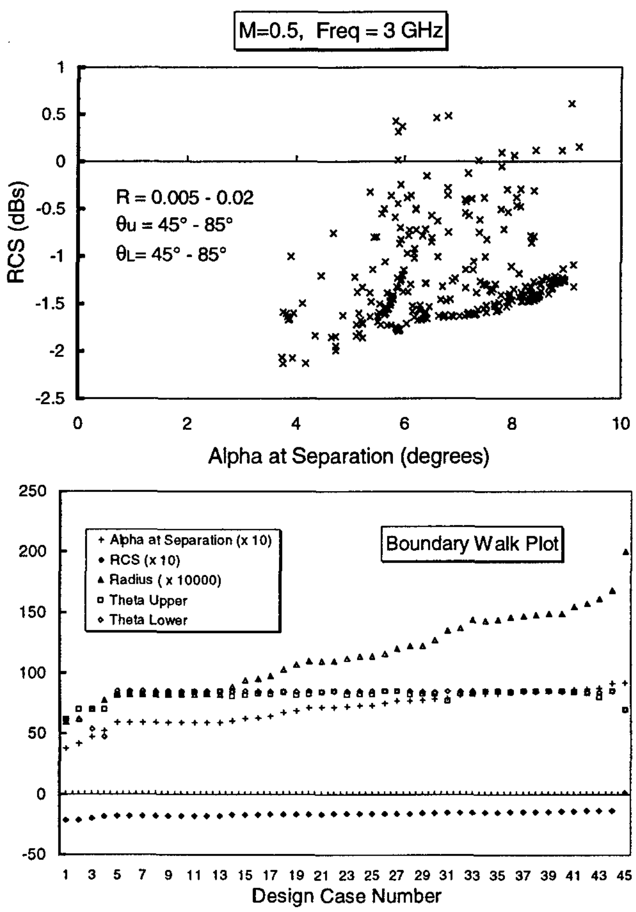


Fig 31 – Broad Range Variables



Extending the allowable range of the design variables results in the fuller boundary of fig. 31, but there is still a clear Pareto boundary. Also shown in fig. 31 are the variations of the design variables along this boundary, with trends reflecting the comments made earlier. In particular the LE radius is allowed to increase to improve the aerodynamic performance without major impact on the RCS. This is since the lower surface angle is at its maximum and since the radar threat is from below.

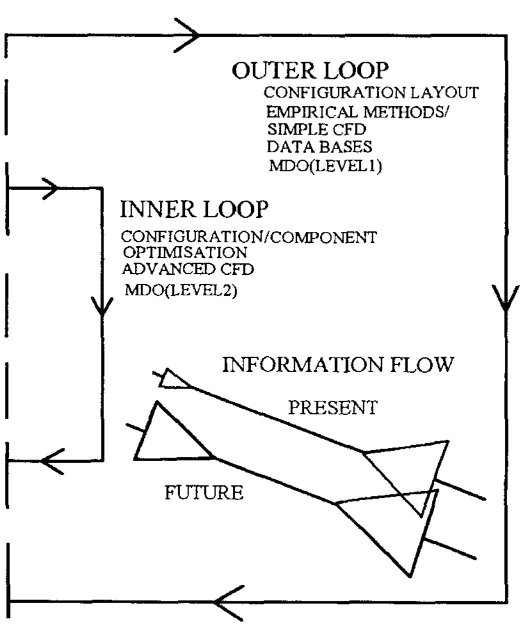
The above simple example has demonstrated the effectiveness of the FRONTIER system to easily obtain trends, exchange rates and Pareto frontiers. However many more evaluations and applications of increasing complexity are necessary to obtain a proper assessment.

7 Future ways of working

The airframe design cycle can be thought of as two design loops as indicated in fig. 32. The outer loop currently adopts empirical and rule based procedures, and dominates the configuration layout and component selection process. The inner, more refined design loop, has a secondary influence in this respect; thus, the controlling information is from the outer to inner loops.

It is anticipated that CFD enhancements, both in terms of physics modelling and faster response times, will enable an immediate feedback to a proposed configuration layout or change, especially on novel configurations where rule based procedures may be less sound. As acceptance of the new design procedures is gained, their use will spread to the outer design region. This will lead to a closer integration and gradual merging of the two design regions.

Fig 32 – Future Ways of Working



8 Conclusions

Optimisation of a combat aircraft is composed of many disciplines, but the aerodynamic database is probably the key component that needs to be in place before the detail optimisation can begin. The choice of the initial configuration is often made based on its high incidence characteristics, thus any optimisation or MDO system needs to capture this component. As much of this is currently outside the scope of CFD there is a strong emphasis in developing a rapid means of obtaining experimental data, in order to bring this element into the MDO environment.

CFD plays a supporting role in the initial configuration study and this will increase as the methods are extensively evaluated and/or the algorithms improve, and a leading role in identifying multi-component interference and further downstream optimisation. The role of numerical optimisation and MDO is currently more sound when viewed in the detail design environment rather than the concept/feasibility stage, for military aircraft.

CFD enhancements, pressure sensitive paint and simplified model build and testing techniques have been highlighted as the means for progressing the design optimisation problem.

9 Acknowledgements

Acknowledgement is given to Mr D. Spicer for undertaking the FRONTIER calculations.

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